98ES-297

# A Summary of the Cassini Spacecraft Thermal Performance from Launch through Early Cruise

**Arturo Avila, Nazilla Rouse, Stuart Clark, Glenn Tsuyuki, and Jerry Millard**
Jet Propulsion Laboratory, California Institute of Technology

## ABSTRACT

Cassini, NASA's mission to investigate the Saturnian system was launched successfully on October 15, 1997. The spacecraft (Orbiter and Titan Probe) is the largest and most sophisticated interplanetary vehicle ever launched. The cruise period from launch until Saturn arrival takes the spacecraft through a wide range of solar/thermal environments (0.67 astronomic units [AU] to 10 AU). The thermal control approach, which consists of thermal design features and operational constraints, must therefore maintain hardware temperature limits throughout this wide range of environments.

The spacecraft nominally points the HGA to the sun so that areas beneath the HGA are shaded while in the inner solar system (<5 AU). The Cassini mission design requires that the spacecraft be able to perform trajectory correction maneuvers with the HGA pointed away from the sun for limited duration's. The off-sun exposure flight experience with interplanetary spacecraft at relatively close heliocentric distance is very limited. Such off-sun maneuvers exposes the nominally shaded spacecraft components to direct solar irradiance. The ability to perform off-sun maneuvers relies heavily on the large thermal capacitance of the spacecraft's central body and the relatively short off-sun duration's required for these maneuvers. An integrated system level thermal balance test was performed prior to launch but off-sun attitude simulation was not feasible because of the size of the spacecraft and cost constraints. The post launch execution of the first trajectory correction maneuver (TCM-1) was the first opportunity to validate the spacecraft off-sun capability and to correct the thermal math model simulation capability.

## INTRODUCTION

SCOPE - This purpose of this paper is to present the spacecraft thermal performance from launch through early cruise (10/15/97 through 2/24/98). This period is characterized by engineering activities, limited instrument maintenance and one TCM. The off-sun exposure flight experience with deep space interplanetary spacecraft at relatively close heliocentric distance is very limited. An off-sun solar characterization was performed in conjunction with TCM-1 when the nominally shaded spacecraft components were exposed to direct solar irradiance. A comparison of flight data with predictions will be presented. Special attention will be focused on the in-flight off-sun maneuvers since ground testing for these maneuvers was not performed. In addition, operational changes resulting from in-flight lessons learned will be discussed.

MISSION DESCRIPTION AND TRAJECTORY - The Cassini spacecraft was launched successfully on October 15, 1997. Since the energy of the Titan IV-B and Centaur launch vehicles was not sufficient for a direct trajectory, planetary gravity assists from Venus (twice), Earth, and Jupiter will enable the spacecraft to reach Saturn by July 2004 (see figure 1). The spacecraft heliocentric distance will vary from 1 AU at launch, to 0.67 AU at the first perihelia, to 10.07 AU at Saturn. During its cruise to Saturn, the three axis stabilized spacecraft will normally point its High Gain Antenna (HGA) towards the Sun. However, during TCM's the spacecraft is turned away from Sun point to accommodate delta V vectors that are not aligned with the solar pointing vector.

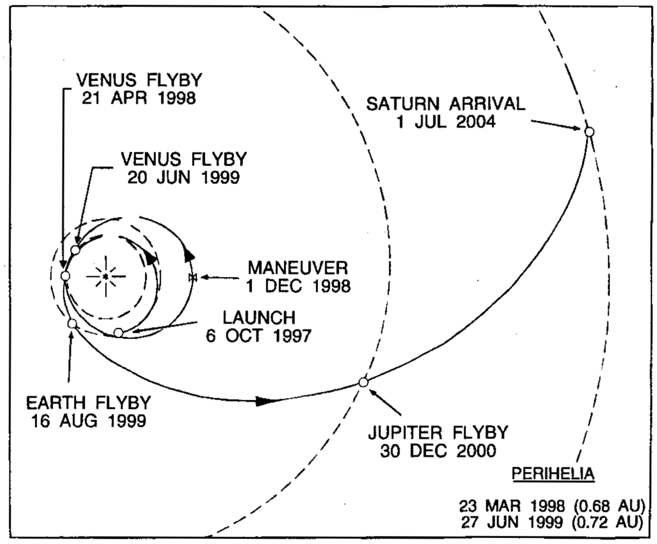


**Figure 1:** Cassini mission trajectory

SPACECRAFT CONFIGURATION - The spacecraft configuration is shown in figure 2. The spacecraft is composed of the Orbiter and the Huygens Probe. The Orbiter was provided by JPL except for the HGA which was provided by the Italian Space Agency and the propulsion module subsystem (PMS) which was provided by Lockheed-Martin. The European Space Agency (ESA) provided the Probe. The engineering systems are mounted throughout the spacecraft, most notably on the BUS and the central body. The most dominant spacecraft feature is the propulsion module central body (PMCB) which is composed of the PMS, upper support structure assembly (USSA) and the lower equipment module (LEM). There are two main engines for redundancy, and during cruise, they are protected from micro-meteoroid damage by a deployable, large, hemispherical cover.

The science instruments are mounted throughout the spacecraft, most notably on the Huygens Probe, the remote sensing pallet (RSP), and the fields and particles pallet (FPP). The dual magnetometers (FGM and VSHM) are located on a deployable boom which is mounted to the BUS. The MAG boom is currently stowed and will not be deployed until after the Earth gravity assist, when the spacecraft will be heading permanently outbound from the Sun. A pivoting dust analyzer (CDA) and a plasma and radio wave instrument are attached to the USSA.

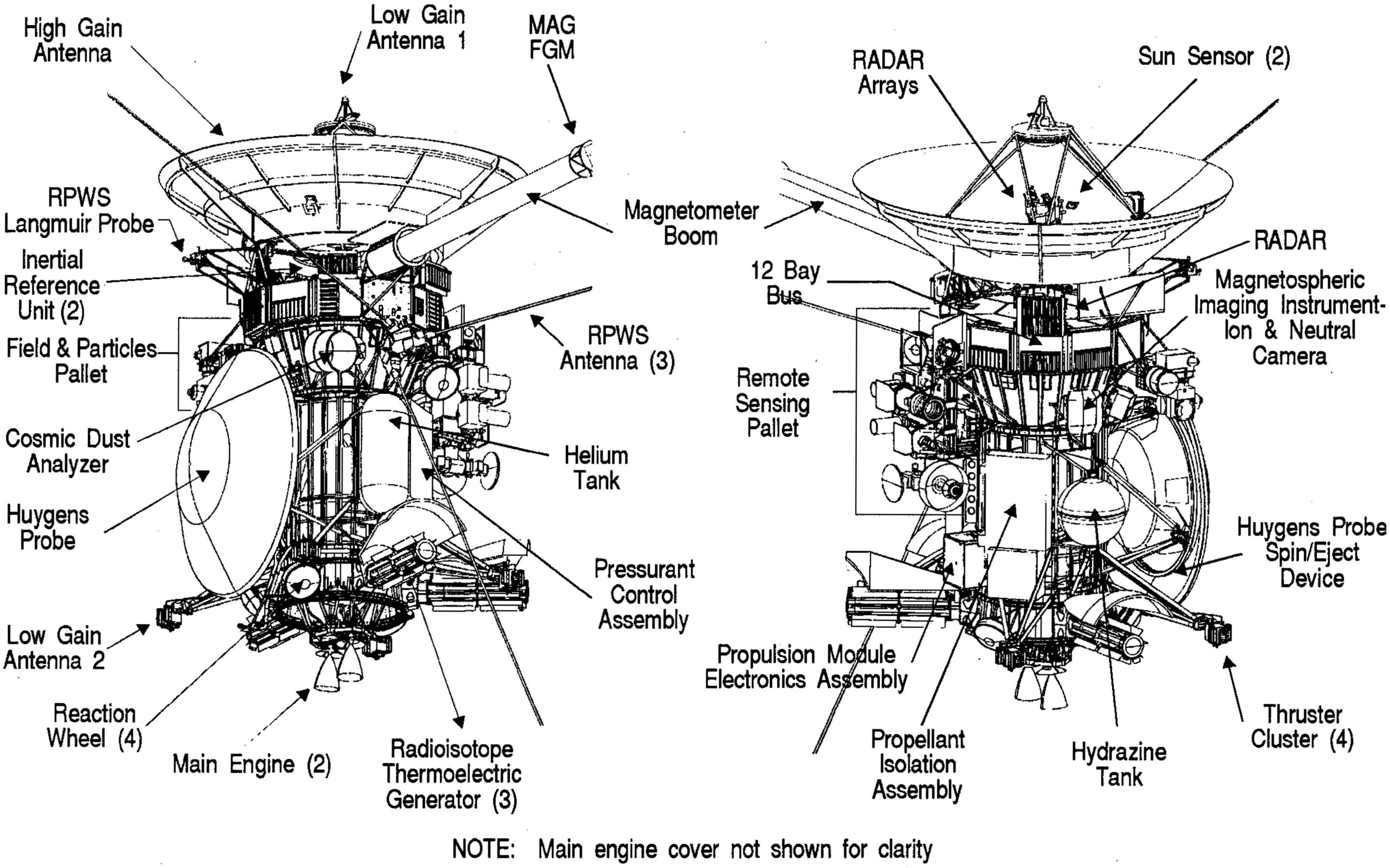



**Figure 2:** Cassini S/C configuration

## SPACECRAFT THERMAL CONTROL DESCRIPTION

THE CHALLENGE - The requirement to satisfy mission objectives at Saturn (10.07 AU) as well as the inner solar system planetary gravity assists (0.67 AU) results in a large variation in heliocentric distance. In order to provide mission trajectory design flexibility (thus optimizing propellant consumption) the spacecraft must tolerate off-sun maneuvers throughout the heliocentric range. The spin stabilized Galileo spacecraft had a trajectory that was comparable to Cassini's but its ability to implement changes to its velocity vector while sun-pointed meant that it did not have to contend with solar exposure due to off-sun maneuvers at small heliocentric distances [1]. The three axes stabilized twin Voyager spacecraft did require off-sun maneuvers but none were required inside of 1.0 AU [2]. The formidable challenge for Cassini was met with thermal design features and operational constraints.

THERMAL DESIGN FEATURES - The spacecraft thermal design features are illustrated in figure 3. The thermal control implementation minimizes the sensitivity to the widely varying environments . The HGA serves as a shade and its structure serves to conductively isolated it from the BUS while the spacecraft is sun pointed. During maneuvers, the Huygens Probe is used as a shade which protects most of the Orbiter's most thermally sensitive hardware [3 and 4].

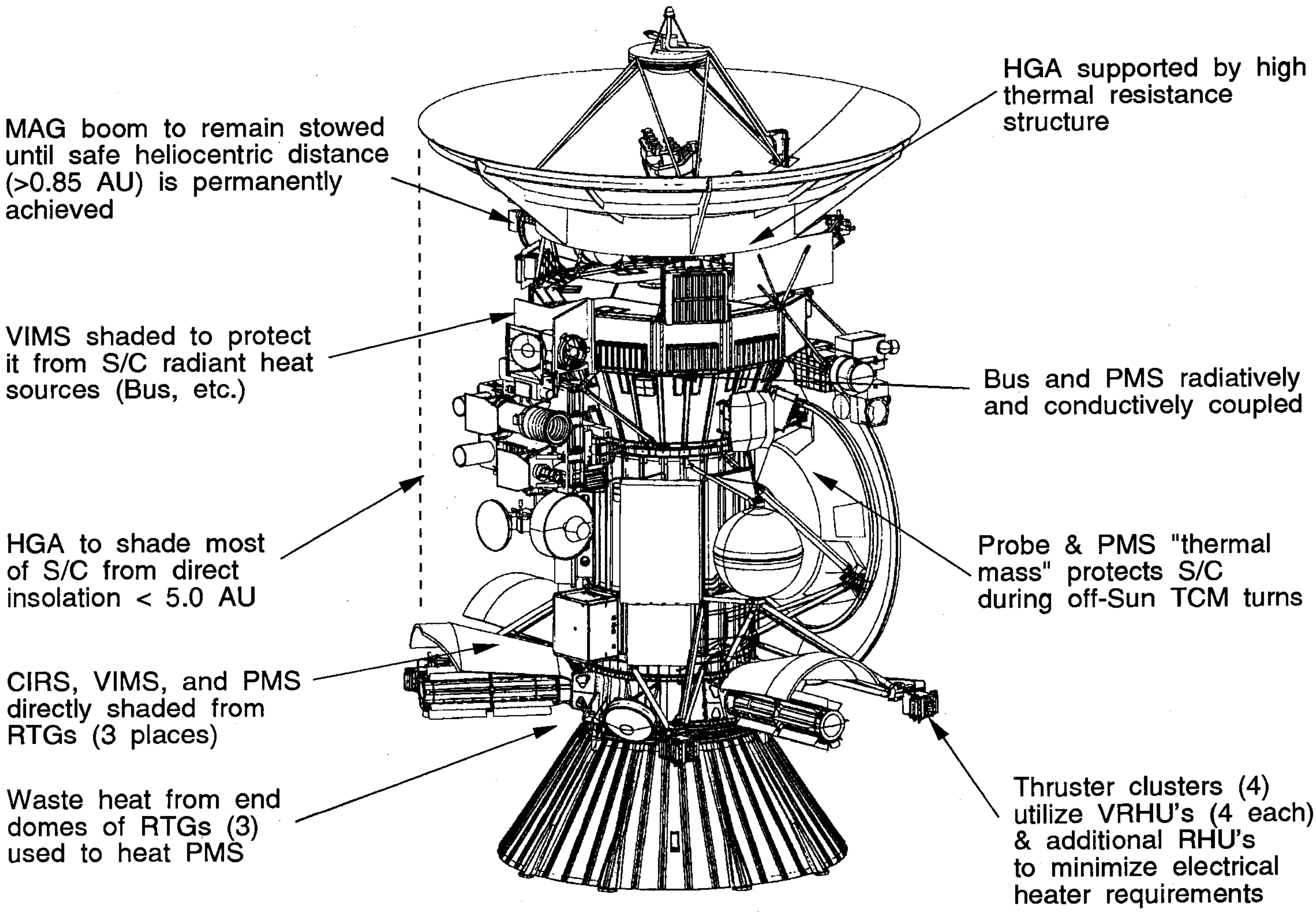



**Figure 3:** System-level thermal design schematic

OPERATIONAL CONSTRAINTS and REQUIREMENTS - The allowable flight temperature (AFT) limits are specified in project documentation [5]. In addition, the AFT documentation specifies how inner solar system off-sun maneuvers should be executed (see figure 4). All maneuvers inside of 5.0 AU are performed in the X-Z plane by turning the -Z axis toward the +X axis to always place the Probe side of the spacecraft in the Sun. The duration of maneuvers inside of 1.0 AU is specified by:

$$\text{Duration} = Y*(AU^2/0.61^2)$$

where Y = 0.61 AU duration for any off-sun angle (30 minutes)
AU = desired maneuver heliocentric distance

The duration of maneuvers between 1.0 AU and 5.0 AU is specified by:

$$\text{Duration} = Y * AU^2$$

where Y = 4 hours if off-sun angle is less than or equal to 60 degrees
= 1.35 hours of the off-sun angle is greater than 60 degrees
AU = desired maneuver heliocentric distance

There are no constraints beyond 5.0 AU. This maneuver strategy protects the most vulnerable assemblies from exposing radiators and apertures to direct solar irradiance. Additional operability constraints, known as flight rules, are also contained in project documentation [6]

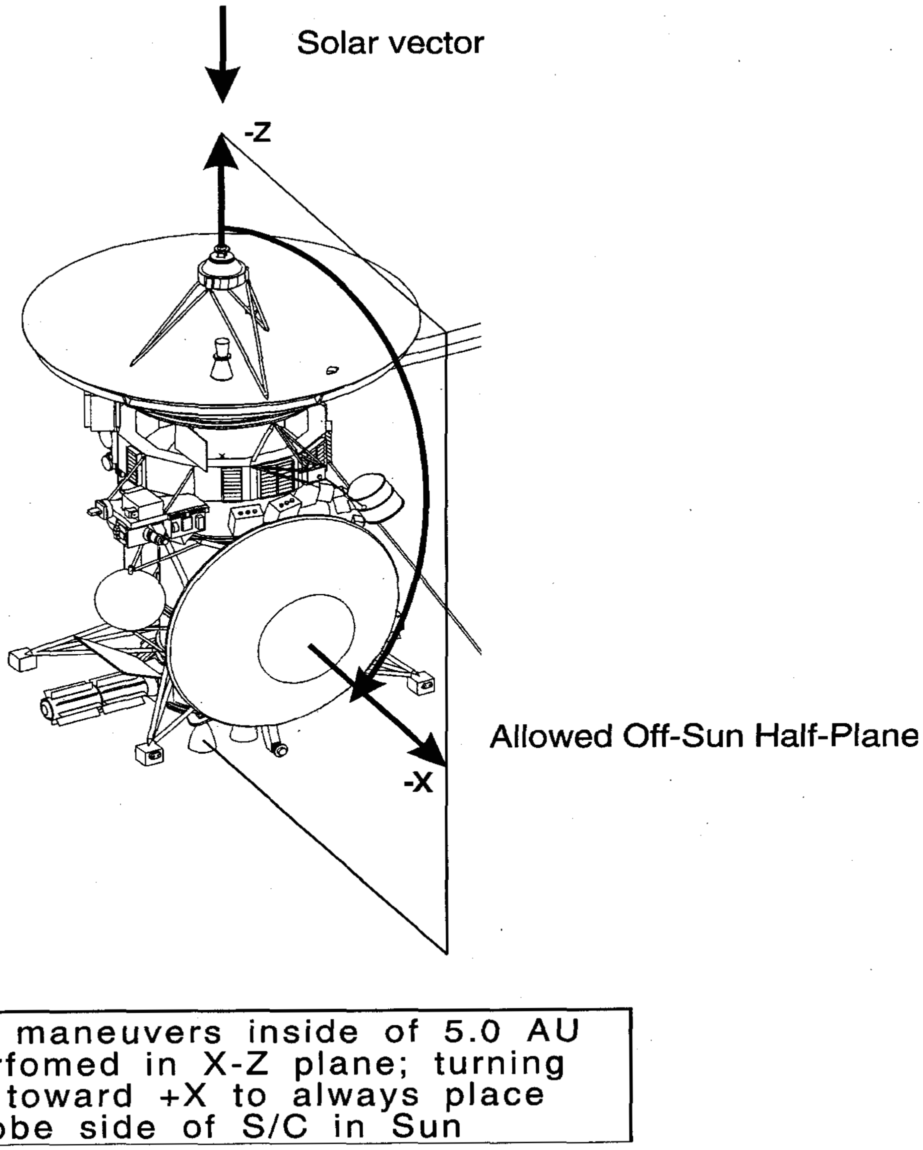



**Figure 4:** Maneuver Execution Requirement

SPACECRAFT SYSTEM LEVEL FAULT PROTECTION - There are three system level fault protection (SFP) algorithms that contribute to the spacecraft's thermal control approach. The first is the Autonomous Thermal Control (ATC) algorithm. ATC is essentially a computer controlled thermostat. The algorithm enables the spacecraft's on-board computers to monitor the temperatures of specified assemblies (up to 12 assemblies) and compare them to on-board thresholds. ATC then responds by issuing a power on or power off command to the assemblies' heater(s) depending on which threshold has been exceeded. The second is the Emergency Overtemperature Algorithm (EOTA). EOTA was implemented to provide some protection against inadvertent off-sun solar exposure. The algorithm enables the spacecraft's on-board computers to monitor the temperatures of specified assemblies (up to 12 assemblies) and compare them to on-board thresholds. The specified assemblies are those that respond quickly to environmental heating and thus provide quick notification of anomalous spacecraft pointing. When EOTA monitors indicate temperatures are exceeding the specified thresholds, the algorithm requests a response from the third SFP algorithm. The third algorithm is called spacecraft SAFING. SAFING sends a request to the attitude control subsystem to point the spacecraft to the sun and also reconfigures the spacecraft to a thermally safe power profile.

## CASSINI SPACECRAFT SYSTEM LEVEL THERMAL MATH MODEL (SCTMM)

SCTMM REQUIREMENTS - The SCTMM was developed as an operational tool for use in mission planning and anomaly reconstruction. The SCTMM was to provide ±5°C agreement for assemblies with relatively small AFT ranges (e.g. BUS Bays) and ±10°C for assemblies with relatively large AFT ranges (e.g. HGA areas and RTG's). The SCTMM simulates, as a function of time, environmental heating, electrical power dissipation, and RTG and RHU thermal decay [7].

SCTMM DEVELOPMENT APPROACH - The SCTMM was developed by reducing and integrating existing subsystem thermal design models. The SCTMM consists of all relevant spacecraft hardware with Space as the only boundary condition. One of the objectives of the Cassini Spacecraft Thermal Balance test was to adjust the SCTMM [3 and 4].

SCTMM POST LAUNCH CORRELATION & OFF-SUN CHARACTERIZATION - The first four weeks after launch, when the trajectory remained at about 1 AU, provided a significant amount of data while at the sun pointed attitude. The design of the first trajectory correction maneuver included having the spacecraft dwell at the delta V vector attitude for the duration specified by the required capability rather than what was needed to achieve the delta V. These data were then used to improve the SCTMM.

## FLIGHT DATA AND PREDICTION COMPARISON

SUN POINTED (HGA TO SUN) ORIENTATION - When the spacecraft is in the HGA to Sun orientation, most assemblies are shaded by the HGA. There are 194 temperature transducers on the spacecraft that are monitored but this paper will focus only on those assemblies that are continuously exposed to solar irradiance (HGA and HGA mounted assemblies) while sun pointed and those that start out in the HGA shade and then become illuminated when maneuvers are executed. A summary of sun pointed flight transducer and SCTMM temperatures for the heliocentric distances where TCM-1 was performed and at the present date (2/24/98), is shown in Table 1. The SCTMM data shown for the 1.01 AU case was generated after model improvements were made following the off-sun thermal characterization. The improved SCTMM was then used to generate predictions for the 0.72 AU case.

**Table 1: Sun Pointed Temperature Comparison**

| Assembly | Requirements Low/High, °C | 1.01 AU Flight, °C | SCTMM, °C | 0.72 AU Flight, °C | SCTMM, °C |
|---|---|---|---|---|---|
| HGA Reflector | -199/125 | -45 | -45 | 10 | 10 |
| HGA X-Bd FSS | -208/129 | 15 | 15 | 73 | 72 |
| LGA-1 | -206/81 | -2 | -4 | 56 | 55 |
| Sun Sensor 1 | -90/80 | -12 | -12 | 28 | 26 |
| Sun Sensor 2 | -90/80 | -10 | -12 | 26 | 24 |
| MAG FGM | -30/80 | 26 | 30 | 31 | 32 |
| MAG VSHM | -30/55 | 8 | 6 | 8 | 6 |
| IRU A | -5/45 | 29 | 31 | 31 | 31 |
| IRU B | -20/45 | 16 | 13 | 18 | 14 |
| BAY 5 | 5/50 | 25 | 26 | 26 | 27 |
| BAY 6 | 5/50 | 21 | 24 | 24 | 25 |
| BAY 7 | 5/50 | 18 | 20 | 21 | 21 |
| BAY 8 | 5/50 | 24 | 24 | 25 | 25 |
| BAY 9 | 5/50 | 22 | 19 | 24 | 21 |
| FPP Structure | N/A | 7 | 8 | 10 | 9 |
| INMS Electronics | -30/60 | 3 | 5 | 5 | 6 |
| INMS Sampling Area | -102/60 | -4 | -4 | -2 | -4 |
| CAPS DPU | -20/40 | -1 | -1 | -3 | 0 |
| CAPS IMSCVR | -20/40 | -6 | -6 | -10 | -6 |
| MIMI CHEMS | -25/40 | 13 | 12 | 15 | 12 |
| MIMI LEMMS ROT | -25/40 | -3 | -3 | 0 | -2 |
| MIMI LEMMS NROT | N/A | 3 | 2 | 5 | 2 |
| CDA EMB | -50/40 | 2 | 0 | -8 | -9 |
| CDA HRD | -30/40 | -7 | -6 | -18 | -19 |
| CDA NROT | -20/40 | 10 | 11 | 7 | 7 |
| RPWS Antenna Assy | -15/60 | 16 | 21 | 17 | 22 |
| PROBE RFE | -20/60 | -6 | -4 | -2 | 0 |
| PROBE PCDU | -40/70 | 11 | 10 | 14 | 13 |
| PROBE Spin Eject Device | -80/70 | -54 | -59 | -50 | -58 |
| Thruster Clusters 1 | 20/60 | 40 | 40 | 41 | 44 |
| Thruster Clusters 2 | 20/60 | 39 | 39 | 41 | 44 |
| Thruster Clusters 3 | 20/60 | 40 | 40 | 42 | 44 |
| Thruster Clusters 4 | 20/60 | 39 | 40 | 40 | 44 |
| REA-A Oxidizer Valve | 5/45 | 33 | 33 | 33 | 35 |
| REA-A Fuel Valve | 0/45 | 17 | 19 | 18 | 21 |
| REA-A Chamber | -1/39 | 5 | 5 | 7 | 7 |
| REA-B Oxidizer Valve | 5/100 | 34 | 32 | 36 | 33 |
| REA-B Fuel Valve | -10/100 | 17 | 18 | 18 | 20 |
| REA-B Chamber | N/A | 5 | 4 | 5 | 5 |

| Assembly | Requirements Low/High, °C | 1.01 AU Flight, °C | SCTMM, °C | 0.72 AU Flight, °C | SCTMM, °C |
|---|---|---|---|---|---|
| LGA-2 | -80/140 | 3 | 3 | 5 | 3 |
| RTG 1 (avg. of 3 sensors) | NA/260 | 248 | 240 | 247 | 247 |
| RTG 2 (avg. of 2 sensors) | NA/260 | 243 | 245 | 243 | 245 |
| RTG 3 (avg. of 3 sensors) | NA/260 | 247 | 240 | 247 | 240 |

OFF-SUN (PROBE TO SUN) ORIENTATION - The spacecraft off-sun maneuver thermal response can be grouped into four classes.

1. Surfaces that are continuously exposed to solar irradiance respond by cooling as the -Z axes turns away from the sun. In some cases the cooling is followed by warming if the off-sun angle is large enough that now the +Z surfaces of HGA mounted assemblies are exposed. Examples include the HGA, LGA-1 and the Sun sensors.
2. Surfaces that are nominally shaded by the HGA, on the -X axis hemisphere, respond by warming as the -X axes turns to the sun. Examples include the BUS Bays and the Fields and Particles Pallet.
3. Surfaces that are nominally shaded by the HGA, on the spacecraft aft end (+Z direction), respond by a combination of warming as the -X axes turns to the sun, the changing power profile, and the main engine burn itself. Examples include the main engine oxidizer and fuel valves and the combustion chamber.
4. Surfaces that remain shaded while sun-pointed and remain shaded during maneuver execution (+X axes) respond only to changes in power profile. Examples include all the assemblies mounted to the RSP. This class of response is not the focus of this paper and will not be presented or discussed.

The total off-sun duration allowed for TCM-1 was 1 hour 22 minutes and 38 seconds at an off-sun angle of 70.6 degrees while at a heliocentric distance of 1.01 A.U. The duration includes the time it takes to turn to and from off-sun attitude (yaw turns). The spacecraft thermal performance during TCM-1 is captured in Figures 5 through 28 (for representative sunlit surfaces). The off-sun flight data clearly indicate that the maneuver approach was sound and no thermal limits were threatened as a result of this activity. Post maneuver correction of the SCTMM yielded acceptable equilibrium and transient agreement. The SCTMM calculated temperature profiles are also shown in Figures 5 through 28.

The class 1 response discussed above can be seen in Figures 5 through 9. Turning off-sun cooled the HGA and LGA1 assembly given that the nominally sun pointed HGA is now viewing deep space and solar exposure is now edge on to the dish. The SSH1 (whose aperture is in the -Z direction and radiator is in the -Z direction) on the -X quadrant first cools as the spacecraft begins the turn off-sun. The SSH1 then warms as the sun sensor radiator, cabling, and MLI wrap are exposed to the sun at this off-sun angle. The SSH2 on the +X quadrant shows only the effect of cooling, since its radiator is shaded by the spacecraft during maneuver execution.

Figure 5
E-2505 HGA Reflector Rear Surface

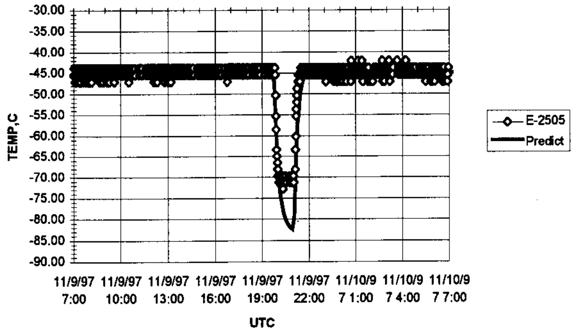



Figure 6
E-2508 HGA X-bd FSS

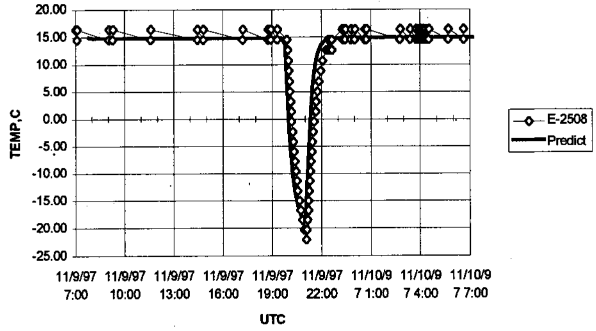



Figure 7
E-2510 LGA-1

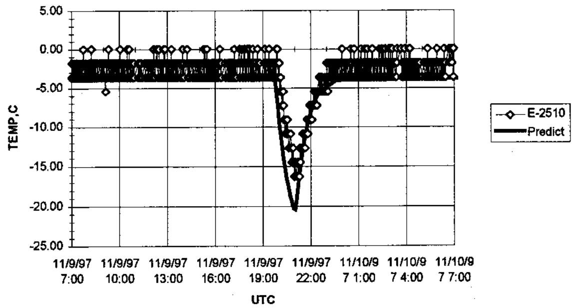



Figure 8
A-0019 Sun Sensor Head 1

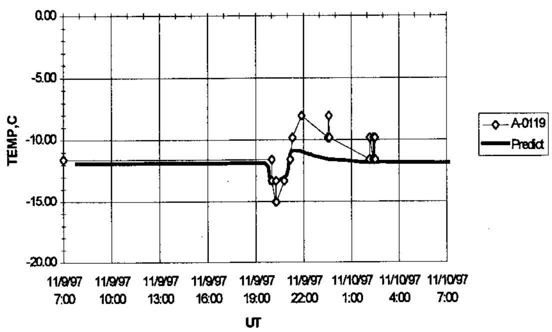
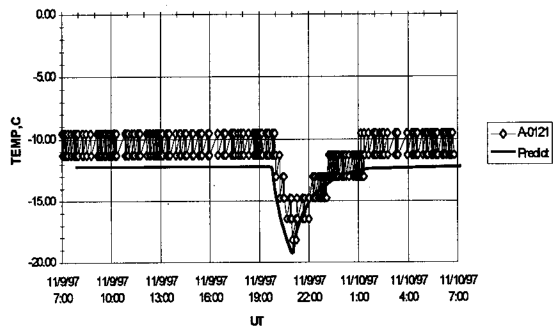



Figure 9
A-0121 Sun Sensor Head



The class 2 response discussed above can be seen in Figures 10 through 23. All of these assemblies are in the -X hemisphere which becomes exposed to the sun during the maneuver. The IRU B (figure 10), Bus bays 5 through 9 (figures 11-15), Probe RFE (figure 16), all show a clear response to solar heating. Solar heating of the fields and particles pallet and the FPP mounted instruments can be seen in figures 17 through 21. The CDA is mounted to the USSA but is also in the -X hemisphere. Its response is shown in Figures 22. The LGA-2 response is shown in figure 23.

Figure 10
A-0105 Inertial Reference Unit B

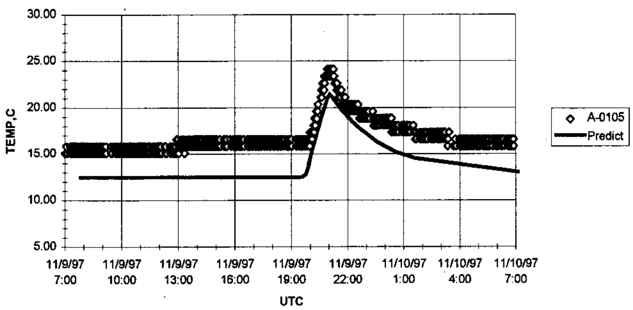



Figure 11
E-2304 BAY 05

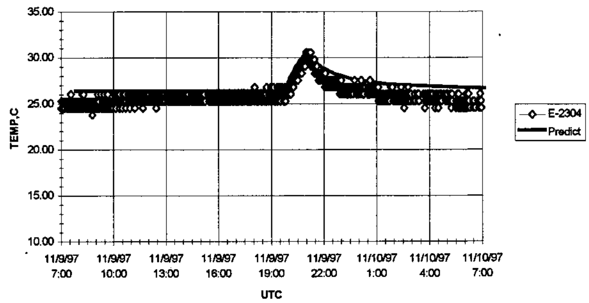



Figure 12
E-2305 BAY 06

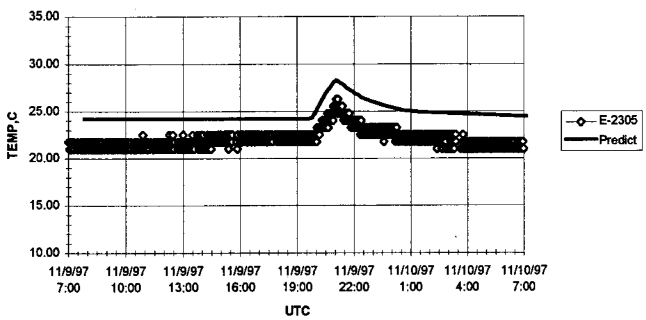



Figure 13
E-2306 BAY 07

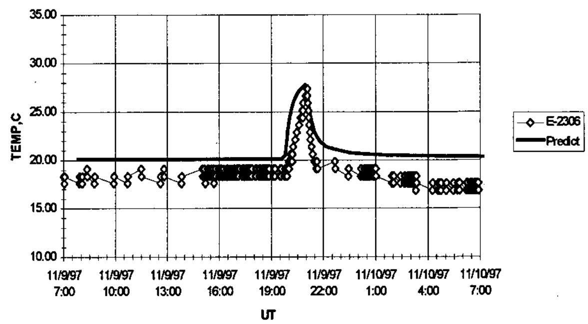



Figure 14
E-2307 BAY 08

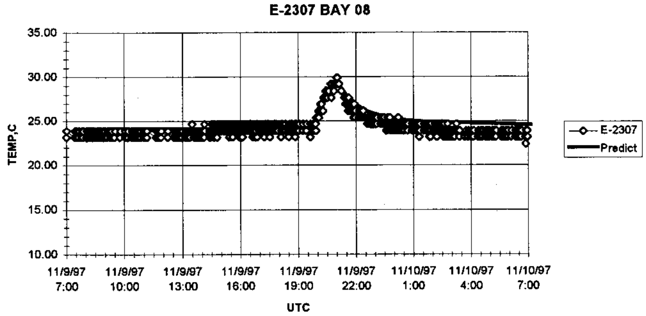



Figure 15
E-2308 BAY 09

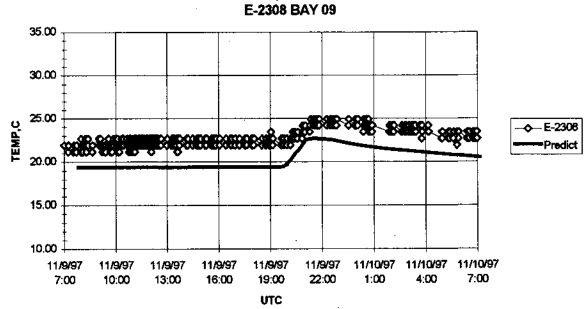


Figure 16
P-3303 Probe RFE

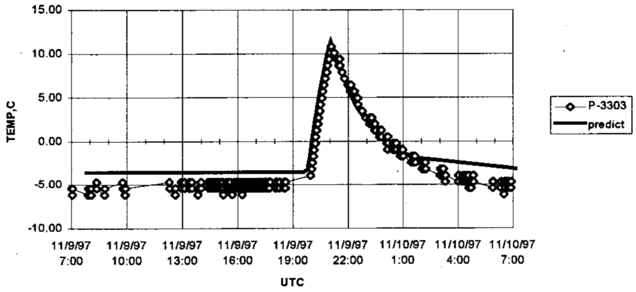



Figure 17
E-0100 Fields & Particles Pallet Structure

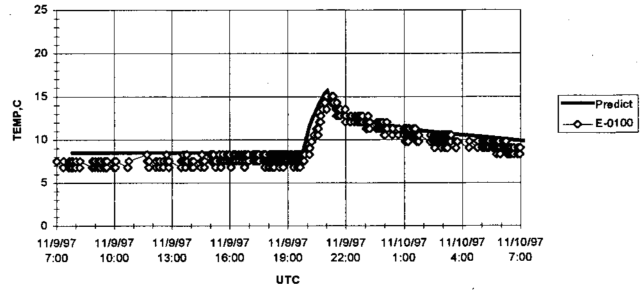



Figure 18
S-1700 INMS Electronics

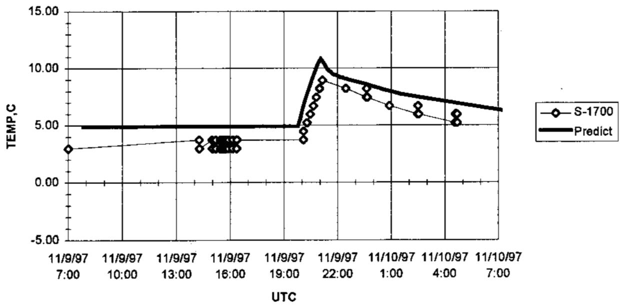



Figure 19
S-2900 CAPS DPU

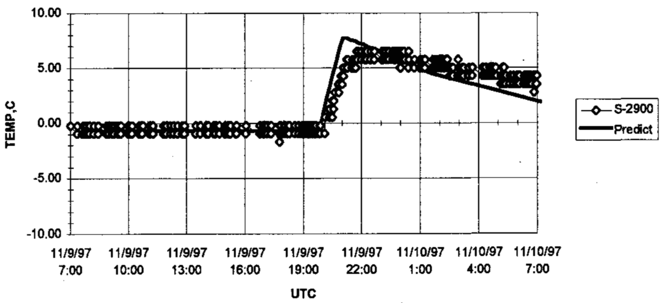



Figure 20
S-2000 MIMI CHEMS

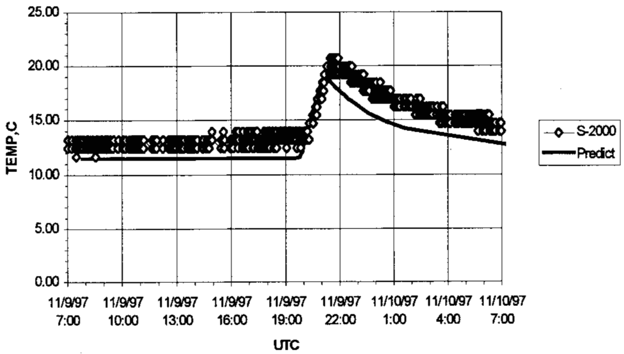



Figure 21
S-2004 LEMMS Stator

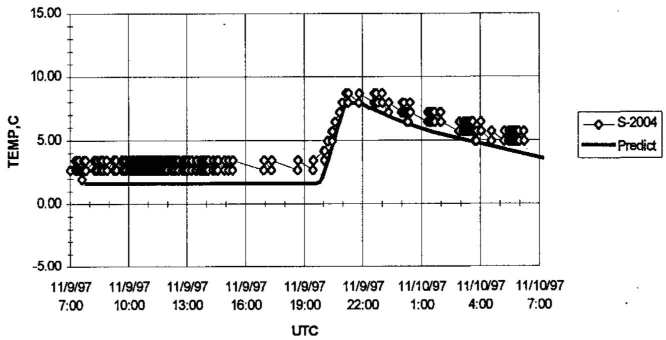



Figure 22
S-2303 CDA HRD T OFF-SUN FLIGHT VS PREDIC

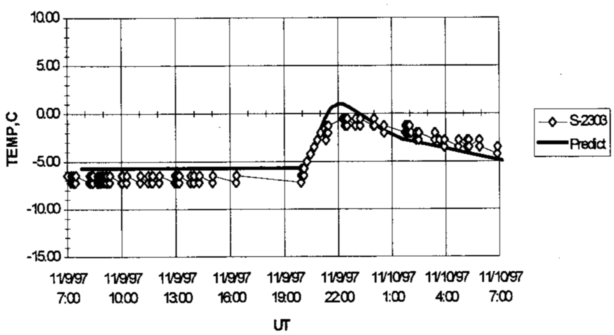
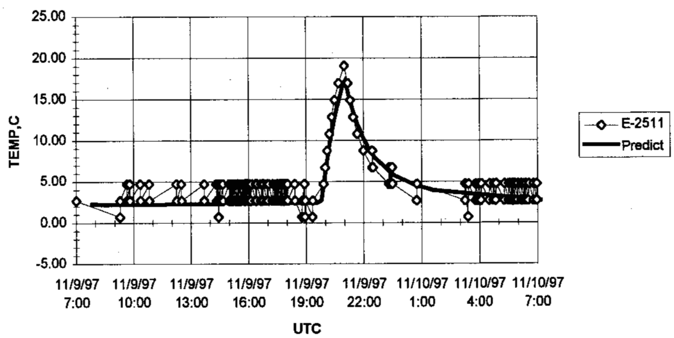



Figure 23
E-2511 LGA-2



The class 3 response discussed above can be seen in Figures 24 through 28. Figures 24 through 25 display the temperature trends on the RCS Thruster Cluster Housings. The trends displayed by Cluster Housings 2 and 3 (in the -X hemisphere) show a somewhat singular temperature spike which includes the thermal response to both turn related burns and solar heating during the off-sun period. The Thruster Cluster Housings 1 and 4 (in the +X hemisphere) were shaded during the TCM-1 maneuver attitude and thus are not included here. Figures 26 through 28 display main engine A temperature trends. All three show a warming of the engine assembly due to the influence of the REA-A engine mounting plate heater that conditions the main engine prior to the burn. Transient spikes in temperature are seen as a result of the main engine burn, being most pronounced for the REA-A Chamber. At the 70.6 degree off-sun angle, the main engine assemblies were shaded by the stowed main engine cover. Solar exposure to the main engine assemblies can be expected for off-sun angles greater than 75 degrees.

Figure 24
E-1756 Thruster Cluster 2

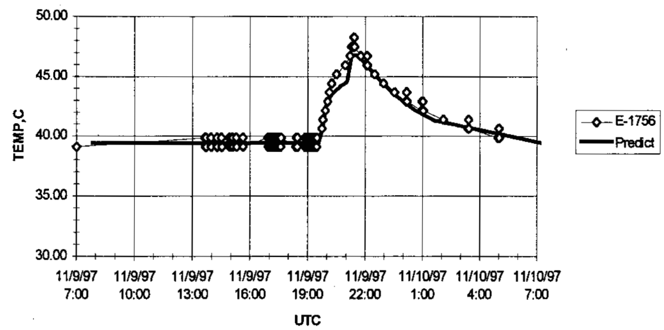



Figure 25
E-1954 Thruster Cluster 3

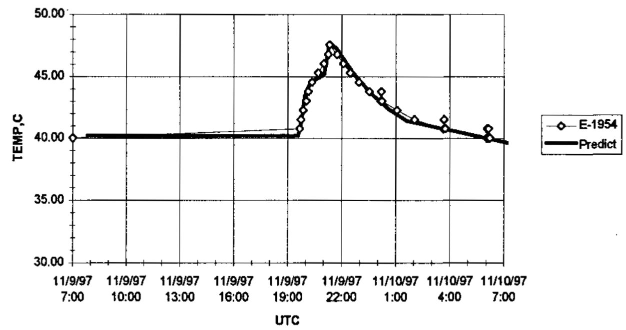



Figure 26
E-1745 REA-A Oxidizer Valve

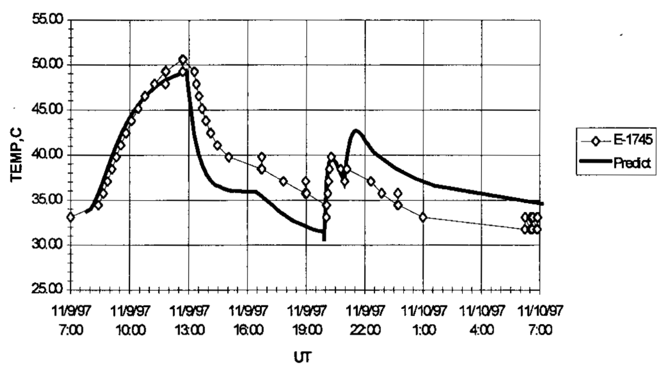



Figure 27
E-1945 REA-A Fuel Valve

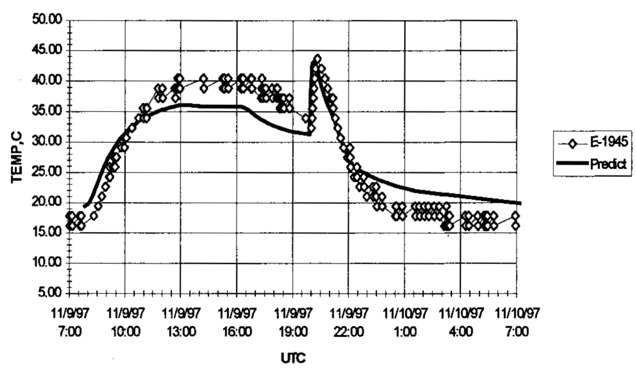
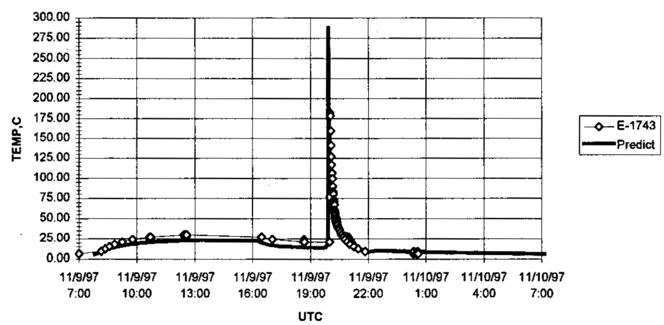



Figure 28
E-1743 REA-A Chamber



The Probe and PMCB are very massive and their transient responses during off-sun maneuvers is very small, if not completely negligible. This fact is, of course, the driving reason for implementing the current maneuver execution approach.

THERMAL ANOMALIES AND LESSONS LEARNED - The only surprise that occurred was the overheating of the main engine A oxidizer valve which occurred while the spacecraft was sun-pointed. The temperature of the main engine A oxidizer valve exceeded it's high "At Ignition" temperature limit of 45°C (figure 26). It peaked at 51°C before a real-time command was sent to turn off the primary oxidizer valve heater. This action corrected the problem and the TCM-1 was completed without further incidents. The problem was due to powering on the main engine mounting plate heater while the oxidizer valve heaters was on. A review of STV data reveals information supporting these events as a nominal response [8]. The scenario tested had the MEA hardware at equilibrium with the MEA Cover open and the PMCB in a worst case cold cruise condition. The test case was meant to conservatively verify that the engine mounting plate heater could elevate the initially cold chamber temperature to its "at ignition" range. This was verified. In addition, the additional power from engine mounting plate heater did not adversely affect all other main engine hardware since the PMCB was in its worst case cold cruise condition with temperatures at the lower end of their requirement range. The STV data indicate that the engine mounting plate heater elevates the oxidizer and fuel valve temperatures approximately 25°C above their initial temperature. This effect was unanticipated but should have been expected based on this data.

The strategy with respect to the use of main engine oxidizer valve heaters and engine mounting plate heaters has been updated and will be used before the next main engine TCM. Associated with this new strategy, Flight Rules that govern the operation of these heaters have been updated.

## CONCLUSIONS

The thermal performance of spacecraft subsystems to date has been exceptional. The only problem (overheating of the main engine A oxidizer valve) that has surfaced is of an operational nature and was subsequently verified as expected response when solar thermal vacuum test data was revisited. A new heater strategy has been implemented to prevent future overheating of a main engine oxidizer valve. This strategy required a change in the use of the main engine oxidizer valve heaters and engine mounting plate heaters. Associated with this new strategy, Flight Rules that govern the operation of these heaters have been updated. The maneuver approach has been validated and comfortable margins are predicted for perihelion conditions.

## ACKNOWLEDGEMENTS

The work described in this paper was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

## REFERENCES

[1] Reeve, R. "Thermal Redesign of the Galileo Spacecraft for a VEEGA Trajectory," *Journal of Spacecraft and Rockets,* Volume 28, Number 2, Pp. 130-138, American Institute of Aeronautics and Astronautics, Washington, D.C., March-April 1991.

[2] Becker, R. "Functional Requirements, Mariner Jupiter/Saturn, Design Criteria for Temperature Control," JPL Internal Document 618-205-3-210, Revision A, November 1976.

[3] Mireles, V. and Tsuyuki, G. "A Summary of the Cassini System-Level Thermal Balance Test: Engineering Subsystems," SAE Paper 97ES-287, July 1997.

[4] Tsuyuki, G., Mireles, V., Lin, E., and Avila, A."A Summary of the Cassini System-Level Thermal Balance Test: Science Instruments," SAE Paper 97ES-287, July 1997.

[5] Tsuyuki, G. "Functional Requirements, Cassini Orbiter, Design Criteria for Temperature Control," JPL Internal Document 699-205-3-210, Revision D, October 1997.

[6] Macias, J. "Flight Rules and Constraints," JPL Internal Document 699-270, Revision C, September 1997.

[7] Avila, A. "Cassini Orbiter System Level Thermal Math Model Requirements," JPL Internal Document D-13846, Revision A, July 1997.

[8] Juanero, K., and Mireles, V., "Cassini Spacecraft System-Level Solar Thermal Vacuum Test Report," JPL Internal Document 699-255, April 1997.